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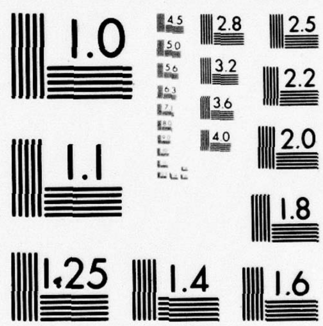

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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Manoeuvre Limitations of Combat Aircraft

NORTH ATLANTIC TREATY ORGANIZATION

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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JOB

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- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;
- Providing assistance to member nations for the purpose of increasing their scientific and technical potential;
- Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community.

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PREFACE

In 1975 the North Atlantic Military Committee requested AGARD to perform a technology study of limitations to the maneuverability of combat aircraft. An improved understanding of the phenomena which limit the maneuverability of combat aircraft and the relationship between these phenomena and the aircraft's physical configuration characteristics, such as geometry, dimensions, mass, mass distribution, control system, etc. will be of significant value in the definition of future military requirements and candidate aircraft designs. This study was to utilize, and not duplicate, past AGARD studies such as AR-82 The Effects of Buffeting and other Transonic Phenomena on Maneuvering Combat Aircraft. The AGARD Steering Committee approved the study and assigned the responsibility for its implementation to the Flight Mechanics Panel.

A Working Group was formally established by the Flight Mechanics Panel in 1976. The Panel recommended that the Working Group utilize the detailed flight and physical characteristics of a large number of current combat aircraft in carrying out the study. Accordingly, the group studied 15 NATO aircraft, and the results of the study have been published in unclassified and classified AGARD reports. In this, the unclassified version, the major results are discussed in general terms. Detailed data on the specific airplanes are presented in the classified report.

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In the period from 1977 to 1978 the Working Group held four meetings and received considerable assistance from advisors and observers who attended one or more of these meetings. A most notable contribution was made by Dennis A. Stangroom, Squadron Leader RAF, Executive - Flight Mechanics Panel, to both the technical and administration activities of the Working Group.

This final report was jointly prepared by the Working Group members. The appreciation of the Working Group is extended to those who prepared the specific combat aircraft characteristics in a consistent and comparable form.

All portions of this report were reviewed in detail by the Working Group members and they concurred in the principal findings.

W. T. Hamilton
Working Group Chairman

CONTENTS

CHAPTER I

INTRODUCTION

Superior maneuverability is a fundamental aircraft performance requirement in the close-in air-to-air and air-to-ground combat environment; however, there are other fundamental military aircraft requirements, i.e. range, payload, safety, procurement cost, operational cost, etc. Consequently the combat aircraft is optimized to best meet the combined requirements and as such it is maneuver limited by different elements or phenomena at different places in its operational envelope:

o Wing Lift - wing size (relative to weight)
- wing geometry (aspect ratio, taper ratio, sweepback, airfoil sections)
- wing high lift and maneuvering system ($C_{L_{MAX}}$, leading-edge system, trailing edge system, maneuver flaps)

o Installed Thrust (relative to drag and weight)

- engine thrust
- inlet and exhaust system effects

o Stability - center of gravity location (relative to aerodynamic center)
- aeroelastic effects (dynamic pressure and Mach effects)
- non linearity (separation, vortices, wake, and shock effects)

o Control limits

- control surface size
- control surface deflection range
- control actuation power
- control aeroelastic effects
- control Mach number effects

o Design Load Limits

- structural strength limits
- pilot limitations
- flutter limitations

o Vibrational Limits

- buffet intensity
- human efficiency
- weapon targeting efficiency

The designer's choice of aircraft weight, wing area, high lift system and installed thrust are, in general, based on desired operating requirements such as range, payload, take-off and landing field length, operating altitude and steady state maneuver capability. The process for choice of these characteristics is generally similar for all configurations. The choice of aircraft detail arrangement and configuration is closely related to desired flight speed, altitude and maneuverability. The maneuver limitations that are directly related to configuration, flight speed and attitude are reasonably independent of airplane size and engine thrust. These limiting flight characteristics include pitchup, wing rock, wing drop, nose slice, and buffeting; this document will concentrate primarily on these configuration-and detail-sensitive limitations and the aircraft characteristics that cause them.

This report is complimentary to AR No. 82 "The Effects of Buffeting and Other Transonic Phenomena on Maneuvering Combat Aircraft" and uses the phenomena definitions as stated in that report.

CHAPTER 2

DISCUSSION OF PHENOMENA

2.0 INTRODUCTION

As previously discussed, the primary objective of the present study was to document various types of physical phenomena which limit the maneuverability of current combat aircraft and to correlate such phenomena for 15 NATO aircraft. Wind-tunnel and flight data were obtained and analyzed for the high-angle-of-attack flight conditions associated with maneuvering flight. Experience has shown that aerodynamic characteristics of airplanes at high-angle-of-attack conditions are extremely nonlinear with respect to static motion variables (angles of attack and sideslip) and dynamic motion variables (angular rates); and the magitude and sense of many important parameters fluctuate rapidly with small increments of angle of attack. Such characteristics are especially prevalent in the lateral-directional axes, as discussed in subsequent sections of this paper. These effects are strongly dependent upon complex aerodynamic flow fields and the particular airplane configuration under consideration. As shown in Figure 2-1, flow conditons at high angles of attack are extremely complex and involve large regions of stalled flow, low-energy wakes from stalled surfaces, strong vortex flows, and aerodynamic interference between various components of the airframe. Basically, it is this poorly defined flow field which produces strong configuration dependency and nonlinear effects.

2.1 PITCHUP

Pitchup, a longitudinal pitch divergence usually experienced at moderate and high angles of attack, can be mild and easily controllable or violent and unrecoverable. In some configurations, the rapid increase of angle of attack caused by pitchup can precipitate various other high angle of attack limitations. Pitchup is the result of an unstable pitching moment variation with angle of attack (or lift). For a conventional airplane configuration with an aft horizontal tail, this unstable variation occurs for only a limited angle of attack region, if at all. Figure 2-2 presents typical lift, drag and pitching moment variations for an airplane with a low, a medium-high, or a T-tail (high) arrangement. The airplane shown with either a medium-high or T-tail arrangement is stable in pitch at both low and extreme angles of attack but unstable at some intermediate angle of attack range. For example, if the medium-high configuration is trimmed (at zero moment) at a low angle of attack, it is stable (a negative slope) when encountering small disturbances; however if the angle of attack is increased by either control movement or vertical gust to where the aircraft is in the unstable region (positive slope) it will either rapidly rotate to the high angle of attack trim point (pitchup) or revert to the low angle of attack trim point. The intermediate zero moment angle of attack trim point is unusable because there exists too great an instability for steady flight with normal manual control procedures.

For some configurations, the high angle of attack trim point can occur at extremely high angles of attack (Figure 2-3), where longitudinal control effectiveness may be severely degraded due to tail stall or impingement of low-energy flow on the horizontal tail. In such a case, recovery from the "deep stall" may be difficult or impossible.

The impact of the deep-stall characteristics on flight motions is shown by the time histories from a piloted-simulator study given in Figure 2-4. On this encounter, the pilot used his usual recovery technique for conventional aircraft (pushing the control column forward). He believed his stall recovery was successful, because the attitude angle came down nearly to the horizontal and the airspeed increased; however, he could not recover from the deep-stall trim condition.

The aerodynamic loads and moments which cause pitchup are the result of airplane wing-fuselage aerodynamic pitching moment variations with angle of attack and the moment contribution of the horizontal tail as the airplane angle of attack is varied. Most swept wing fighters have a wing separation or gradual stalling characteristic in which the aft outer part of the wing stalls progressively as the angle of attack is increased (Figure 2-5). The unseparated portions of the wing and fuselage continue to increase in lift with angle of attack while the separated portions have a constant or even decreasing lift contribution with increasing angle of attack. Because the wing airflow separation is initiated near the wing trailing edge and at the mid or outer portion of the wing and grows rapidly in the outer (most rearward) portion of the wing, it automatically causes an additional unstable moment with increasing angle of attack until the wing approaches a completely stalled condition. An approximate wing-fuselage pitchup boundary is presented in Figure 2-6. These boundaries have been empirically derived from the characteristics of past aircraft.

The horizontal tail contribution to airplane stability can either aggravate or minimize a basic wing-fuselage pitchup characteristic depending upon its area, span and location relative to the wing. The tail contribution to stability is dependent upon the airflow downwash variation it encounters as airplane angle of attack is increased. Figure 2-5 illustrates an airplane with a swept wing and three tail positions. It shows a typical inward and upward movement of the rolled up wing vortex core, as angle of attack is increased beyond that at which wing separation is initiated. The larger and heavier vortex indications show the stronger vortices associated with the higher lift values resulting from higher angles of attack. The general downwash contribution to the tail stability increment increases with vortex strength (or lift coefficient) and its location (both horizontal and vertical) relative to the tail. The tail contribution to stability will aggravate the tail-off pitchup characteristics (increase the degree of instability) until the vortex core moves above the horizontal tail plane. A horizontal tail located near or slightly below the wing chord plane will contribute an increasing stability increment with angle of attack and can reduce or eliminate the wing-fuselage pitchup characteristic.

Pitchup characteristics often occur at high subsonic or transonic Mach numbers where relatively strong shockwaves are located onthe wing upper surface. With relatively small changes in angle of attack, the shockwave can change in strength and location, thereby affecting the degree and location of any wing separation, which in turn affects both the wing-fuselage stability characteristics and the downwash characteristics in the region of the tail. Airplane pitching moment (and pitchup) characteristics are extremely configuration sensitive. The only reasonably accurate indication of an airplane's pitchup characteristics (or lack of

pitchup) prior to flight testing comes from wind tunnel tests of an accurate model carried out at flight Mach numbers and at Reynolds numbers in excess of one or two million.

Modern fighters are designed against pitch-up from the outset, and consequently the violent type pitch-up tends to disappear fromthe current fighter inventory. However, with highly maneuverable fighters a special type of pitch-up requires a careful consideration. This situation exists when sideslip generates a nose-up pitching moment (Figure 2-7). In an aggressive maneuver this characteristic, when combined with inertia coupling and (or) with negative stability, can produce a sudden and very dangerous departure. As illustrated in the figure, a configuration which can be trimmed to α_3 = point A at zero sideslip can only be trimmed to α_1 = point 3 (much lower than α_3) under conditions of sideslip and inertia coupling.

2.2 WING ROCK

Wing rock is an uncommanded rolling motion which varies greatly in terms of the physical characteristics of the motion experienced. At transonic speeds, the motions are usually pure rolling motions or rolling and yawing motions, which may be generally random and nonperiodic. These particular motions become a tracking and tactical maneuver limit if roll rates greater than about 10°/sec are experienced, or if significant yawing motions are present. At subsonic speeds, wing rock is usually characterized by periodic, large-amplitude rolling and/or yawing motions which may be self-sustaining and limit cycle in nature; or the motions may be typical of a linearly stable or unstable system. For subsonic flight conditions, wing rock constitutes a maneuver limitation if the angle of attack region associated with wing rock occurs in the tracking angle of attack region, significantly below maximum lift. On the other hand, wing rock for some configurations serves as an inherent form of stall warning if it preceeds a more serious characteristic, such as nose-slice or pitchup. If, in this case, the wing rock is not divergent, it is usually acceptable to the pilot. Finally, in the approach and landing condition, wing rock can be extremely dangerous from a safety of flight standpoint in view of thehigh roll rates (as high as 50°/sec) and large bank angles (±90°) which may be encountered.

In addition to the tracking limitations imposed by wing rock, additional tactical and flight safety problems may be experienced. For example, wing rock may be easily excited by lateral control inputs, and severe pilot-induced oscillations can be experienced, resulting in controllability problems ranging from degradation of tactical effectiveness during combat maneuvers to flight safety during landing approach. Also, the existence of wing rock must be considered during the design of advanced flight control systems for enhancement of flying qualities and spin resistance at high angles of attack. For example, use of a high-gain aileron-to-rudder interconnect for elimination of adverse yaw at high angles of attack can aggravate and amplify the wing rock motions to the extent that the airplane/pilot combination becomes tactically ineffective; and the interconnect system must be drastically modified or eliminated.

A flight test timehistory of a current fighter configuration encountering wing rock during a tracking-task is presented in Figure 2-8. The pilot initiated the maneuver by banking the aircraft into a left turn and increasing g loading by pulling back on the control stick while avoiding intentional lateral stick inputs. As angle of attack increased, the airplane suddenly exhibited large-amplitude wing rock which completely negated the tracking capability of the pilot. In order to regain tracking, the pilot reduced angle of attack andhad to reacquire his target.

The seriousness of the degradation in tracking which may be experienced due to wing rock is illustrated in Figure 2-9. Data are presented which indicate representative values of C_L for onset on buffet and wing rock for a recent fighter configuration. Data are also presented indicating the relative magnitude of normal accelerometer readings of buffet as measured at a wing tip station. As shown at the right side of Figure 2-9, the effect of buffet on miss distance during tracking was noticeable, although relatively small. However, when the aircraft encountered wing rock, the tracking error became very large, and the pilot was only able to track for brief periods of time.

Considerable research has been conducted in the U.S. and Europe on the identification and aerodynamic causes of wing rock. Wind-tunnel tests and flight studies at subsonic and transonic speeds have indicated several potential causes of wing rock, as indicated in Figure 2-10. At the present time, additional research is believed to be needed to verify that aerodynamic hysteresis is a major factor causing wing rock; but there is little doubt that aerodynamic nonlinearities, shock-induced separation at transonic speeds, and loss of roll damping near wing stall are major factors.

At subsonic speeds, it appears that the major factor producing wing rock is loss of roll damping. some results of wind-tunnel forced-oscillation tests of a current fighter with wing rock are presented in Figure 2-11. The data show that the magnitude of the damping-in-roll parameter, $C_{l_p} + C_{l_{\dot\beta}} \sin \alpha$, decreased markedly at angles of attack near wing stall, resulting in a loss of roll damping near $\alpha = 12°$. The magnitude of the effective dihedral derivative C_{l_β}, however, remained large near $\alpha = 12°$. As a result of these aerodynamic trends, the aircraft exhibited wing rock in the angle of attack region indicated.

Experience has shown that certain airframe modifications, such as wing-fuselage strakes, externally-mounted wing tanks, and wing leading edge flaps or slats can have significant effects on roll damping and wing rock. In some cases, use of combat slats has resulted in aggravation of wing rock.

Although the foregoing loss of roll damping can be identified as a cause of wing rock, the mathematical modeling of such aerodynamic behavior normally requires additional data for valid simulation of fighters at high angles of attack. In particular, wind-tunnel studies have shown that the effects of frequency and amplitude of the oscillatory motions experienced during wing rock can have very large effects on the magnitude of damping in roll. For example, shown in Figure 2-12 are results of forced oscillation tests over a range of reduced frequency and oscillation amplitude for a fighter configuration at $\alpha = 30°$. The large effects of frequency and amplitude would be expected to be extremely important in the calculation and theoretical analysis of wing rock. It has been necessary in most analytical and simulator studies to obtain similar data for valid simulation results, and it is extremely important to recognize the existance of such effects in studies of high angle of attack flight dynamics.

2.3 WING DROP

The definition of wing drop given in AGARD-AR-82, Section 1.5.3 is "Wing drop (or "Roll off"): This is an uncommanded motion seen by the pilot as a divergence in roll and an incipient departure. Typically the roll rates are not high, being of the order of 10-20°/sec. It is clearly beyond both the aiming limit and the tactical maneuvering limit, and immediate recovery action is required in order to maintain full control."

A flight record which shows evidence of wing drop at transonic speeds is shown in Figure 2-13. It occurs at about 1.2 seconds, when there is no aileron or rudder activity, but angle of attack is being increased. The pilot did not attempt to cancel the roll rate, as he was endeavoring to obtain a wing rock response with minimal control inputs.

The dynamic response is presumably due to rolling moments induced by asymmetric flow on the wings, which may be caused by either "premature" flow separation on one wing, or by shedding of asymmetric vortices from the fuselage nose, although the latter usually have more significant effects in yaw. Rolling moments at zero sideslip are sometimes apparent in wind-tunnel tests, and are usually attributed to small asymmetries in the geometry of the model if they are significant at low angles of attack. However, a sudden change in level of rolling moment can occur at higher angles of attack (see Figures 2-14 and 2-15), particularly at high subsonic Mach numbers. These changes can often be measured consistently (i.e. are not random) and are maintained for small sideslips. The conditions for flow separation are obviously very sensitive to local imperfections in the model or in the tunnel flow, to Reynolds number, and possibly to vibration of the model, and so extrapolation to full-scale is difficult. In addition, definite asymmetries are introduced on actual aircraft, via pilot heads, vanes, aerials etc., while unintentional geometric asymmetries are likely to be particular to individual aircraft. Use of ailerons and/or spoilers obviously affects the flow over the wing asymmetrically, so that the dynamic situation in flight is likely to differ from the static tunnel test conditions.

However, it is of interest to compare the magnitudes of the rolling moments measured in a wind tunnel, with the step change required to give an initial acceleration in roll of, say, 1 rad/sec^2, using

$$\dot{p} = \tfrac{1}{2}\rho V^2 Sb\, C_\ell / I_x$$

For flight at 20000 feet,

$$C_\ell = \frac{I_x}{32600\, M^2 Sb}\,\dot{p} \quad \text{(SI units)}$$

and the measured changes in C_ℓ are seen to be sufficiently large to cause significant roll acceleration (Figures 2-14 and 2-15). The value of $10^3 C_\ell$ needed to give $\dot{p} = 1$ rad/sec^2 is shown by the length of the vertical bar, which is drawn at the angle of attack corresponding to the maneuver boundary at the Mach number, for each aircraft. The first three examples, Figure 2-14 are for aircraft whose maneuver boundaries are defined by onset of wing rock and/or wing drop. Wind tunnel results in Figure 2-14(a) indicate large changes in level of rolling moment, which are sensitive to flow conditions near the leading-edge, shown by the effects of transition bands. The changes would cause significant initial rolling response, but they usually occur at lower angles of attack than indicated by the flight maneuver boundary. This is consistent with the effect of Reynolds number, flow separation being delayed to higher angles of attack as Re is increased to flight values. There is possibly closer correlation between flight and tunnel results in Figure 2-14(b) where the tunnel Reynolds number was approaching that of the flight tests. The amplitude and extent of the oscillatory C_ℓ shown in Figure 2-14(c) depended on the vortex generators and fences arrays, and flight behavior of aircraft with with the devices at corresponding positions (dependent on the difference in Reynolds number), seemed to correlate qualitatively, i.e. roll unsteadiness for oscillatory tunnel C_ℓ, and wing drop for change in level of tunnel C_ℓ.

Tunnel measurements of $C_\ell(\alpha)$ at $\beta = 0°$ are also available for three aircraft which have acceptable roll characteristics in flight (Figure 2-15), but unfortunately they show similar changes in level to the results discussed above, at angles of attack below the maneuver boundary. There may be acceptable reasons for this, in that the Reynolds number of the tests shown in Figure 2-15(a) was low (1×10^6), and the results in Figure 2-15(c) are for zero leading-edge flap setting, which does not correspond to the scheduled flap settings used in flight.

Thus it does not seem possible to find convincing correlation between tunnel measurements and flight behavior, although $C_\ell(\alpha)$ at $\beta = 0°$ may give some qualitative indication of wing drop tendency. The resulting response depends on many other factors, such as rate of increase of angle of attack, pilot reaction, damping-in-roll etc., and so specific design criteria cannot be defined.

In summary, the features likely to influence the magitude of the rolling moments at zero sideslip are:

(a) wing design, influencing the type of flow separation as asymmetric loads due to flow near the tip separating first have larger moment arms than changes to loading inboard;

(b) nose shape, which may be designed to reduce the magnitude of asymmetric vortices;

(c) devices on the wing to control flow separation, as these are designed to produce definite conditions for separation, rather than the random "free" separations on a clean surface.

Factors affecting the initial response of the aircraft to rolling moments are:

(a) inertia ratio, as aircraft with low inertia in roll are sensitive to roll disturbances;

(b) damping-in-roll derivative, or damping of roll subsidence mode;

(c) effectiveness of stability augmentation system.

2.4 NOSE SLICE

AGARD-AR-82, Section 1.5.4 defines Nose Slice (or "yaw off") as a divergence in yaw and an incipient departure. No aiming is possible after its onset, and indeed, in contemporary aircraft, by the time the pilot has recognized the symptoms, it is usually too late to prevent the incipient spin departure. This and wing drop are typical of the motions resulting when the pilot pulls back on the stick to get that last bit of turning performance out of the aircraft.

A typical directional divergence as illustrated by the flight time histories presented in Figure 2-16. This figure presents flight recorder traces of the principal flight variables during an accelerated stall at 7620 m (25 000 ft.) with the airplane configured for cruise flight ($M = 0.4$). The maneuver was initiated by rolling to a 60° banked turn to the left and then increasing the angle of attack at an approximately constant rate. As the angle of attack was increased, lightly damped wing rock became noticeable. At about 44 seconds severe wing rock was experienced; at about 50 seconds the oscillation diverged violently and the airplane entered a 2-1/2 turn spin to the right.

With the advent of swept-wing, close-coupled jet fighter designs, nose slice has become a common phenomenon, and considerable research has been conducted regarding the physical causes and potential cures. Yaw divergence at high angles of attack can be caused by three primary factors: loss of directional stability, asymmetric yawing moments, and adverse yaw. Such factors may be produced by extremely nonlinear aerodynamic phenomena, and they are usually configuration dependent to the extent that small variations in configuration details may result in a dramatic improvement or degradation in characteristics. As a result of this configuration dependence, it is very difficult to generalize the data presented herein for current fighters.

An indication of the factors associated with nose slice caused by loss of lateral-directional stability at high α is provided in Figures 2-17 and 2-18, which show wind-tunnel data for the F-4 configuration. The variations of the static lateral-directional force and moment coefficients with angle of sideslip for angles of attack of 0° and 25° are presented in Figure 2-17. As can be seen, the variations of the lateral-directional coefficients with sideslip become very nonlinear at $\alpha = 25°$, and the concept of stability derivatives for such conditions must include a consideration of the range of sideslip involved. For analysis of stability characteristics, the data are usually analyzed over a sideslip range of about $\pm 5°$. Stability derivatives obtained for the present configuration for sideslip angles of $\pm 5°$ are summarized in Figure 2-18. The data were obtained at Reynolds numbers of 0.5×10^6 and 4.3×10^6. The two sets of data agree fairly well, with the exception of a slightly lower level of directional stability for the higher Reynolds number at moderate angles of attack. Both sets of data indicate a marked decrease in $C_{n\beta}$ as angle of attack is increased, $C_{n\beta}$ being negative at angles of attack above 22°. The data also indicate that as the angle of attack exceeds 15°, a substantial reduction in effective dihedral occurs.

The use of pointed fuselage forebodies for present-day supersonic aircraft can have large effects on the stability and control characteristics of these vehicles at high angles of attack. These shapes have been found to produce large asymmetric yawing moments which can be much larger than the corrective moments produced by deflection of a conventional rudder. These moments may have a predominant effect on stall and spin characteristics and can, in fact, determine the ease and direction in which an airplane may spin.

The results of many investigations have indicated that the large asymmetric yawing moments produced by long, pointed fuselage forebodies are caused by asymmetric shedding of vortex sheets from the nose. As shown in Figure 2-19, flow separation on a long nose at zero sideslip tends to produce a symmetrical pattern of vortex sheets at low angles of attack. This symmetrical flow pattern does not produce any side force on the nose; consequently, no yawing moment is produced. At higher angles of attack, however, the vortices increase in strength; the flow pattern becomes asymmetrical; and the asymmetrical flow produces a side force on the nose which, in turn, produces a yawing moment about the airplane center of gravity.

Wind tunnel tests have frequently revealed such side forces and yawing moments at zero sideslip which were attributed to model asymmetries and ignored in the data analysis. Only recently have flight test results confirmed the existence of these forces and moments at high angles of attack. Figure 2-20 shows a comparison of yawing moment coefficient at zero sideslip between a small scale wind tunnel model of the F-5 airplane and the full scale vehicle. These results confirm the accuracy of the wind tunnel prediction. Very fine increments in pitch, of the order of one degree, must be used in the wind tunnel to define the true character of the variability with angle of attack.

As previously discussed, the nose-slice phenomenon is extremely dependent on configuration details, to the extent that wind-tunnel studies are required to identify the existence of the problem. Several concepts have been identified to "soften" or eliminate nose slice, including: leading-edge slats, vertical tail location, and fuselage forebody shaping.

High angle-of-attack directional stability can be profoundly influenced by the cross sectional shape of the fuselage forebody. Figure 2-21 illustrates some of these nose shape effects on a wind-tunnel model with no vertical tail. All configurations are unstable at low angles of attack, as would be expected. At an angle-of-attack of approximately 25 degrees, the effects of cross section shape become evident, with the vertical ellipse becoming more unstable and the horizontal ellipse breaking in a stable direction. These data lead to the conclusion that a flat elliptic nose is desirable for directional stability at high angles of attack.

Small, thin strakes along the maximum half-bredth of the nose are effective in stabilizing the forebody vortex system, preventing the occurrence of the previously discussed asymmetric flow pattern at zero sideslip. Figure 2-22 shows the reduction in magnitude of the peak yawing moment at zero sideslip obtained with nose tip strakes in the radome region. The effects of such strakes on other characteristics such as directional stability, pitching moments at sideslip, and radar performance must also be evaluated.

2.5 BUFFETING

An extensive discussion of the effects of buffeting on combat aircraft was the task of a special AGARD-FMP Working Group, the results of which are summarized in AGARD-Advisory Report Nc. 82. Within that report are discussed: definitions and criteria, analysis-, wing-tunnel- and flight test methods, the influence of configuration factors, and possibilities for the improvement of buffet characteristics. Within the present report only some general important aspects will be repeated from the previous report. Additionally, a summary of the buffet data of the aircraft presented in this report and the results of an attempt to find correlations between buffet and geometrical parameters will be shown.

The following definition of buffeting from a pilot's point of view was given in AGARD AR-82: "Buffeting is a vibration which is perceptible to thepilot to a degree that intrudes into his concentration on his maneuvering task and may interfere with the precision of his control.

It is not important from the handling standpoint whether it comes from wing flow separation, separated flow striking part of the airframe, intake flow breakdown, stores interference, spoilers, airbrakes, bomb doors, or other devices that change the shape of the aircraft. It may, of course, be significant if the part of the structure that is vibrating is that where the pilot or the weapon is situated.

To the fighter pilot who knows his aircraft, buffet onset is a valuable source of information in moments of intense activity when he is not able to refer to his flight instruments. Of the many different buffet level criteria to be found, the following is a summary which smooths out the variations. The "g" values quoted are peak values.

Onset	$\pm$ .035	to	.1 g_z	perception depends on workload/normal	g
Light	$\pm$ .1	to	.2 g_z	definitely perceptible	
Moderate	$\pm$ .2	to	.6 g_z	annoying	
Severe	$\pm$ .6	to	1.0 g_z	intolerable for more than a few seconds	

Provided that there are no other effects such as loss of full control or random aircraft motions, light buffet usually has no adverse effect on maneuvering, either coarsely or precisely. The average fighter pilot is so used to flying in this region that he may not even comment on it at the lower amplitudes. He will however feel annoyance and frustration when the buffet characteristics reach the level where his ability to track his target is affected; other effects on his performance may result fromthe arm mass feedback to the stick and his ability to see the target or his cockpit controls and instruments. At the intolerable level the motion becomes physically punishing, and full control is not possible as a result of the effect of the buffet on the pilot himself.

The significance of buffet in air combat depends upon the task. If flight in buffet gives a performance improvement then pilots will use this region during the tactical phase of combat. Tracking will also take place at quite high buffet levels, even with guns; but when the low frequency, high amplitude "bouncing" buffet occurs then there is no further advantage to be gained from operating in this region."

Based on various studies, including the findings of AGARD AR-82, it can be definitely stated that buffet onset and even a significant increase in buffet intensity does not deter a fighter pilot to penetrate deeply in the buffet range to attain a firing position. Buffet can be a limiting factor in the offensive situation only when the airplane disturbances reach a level where aiming accuracy is seriously degraded (low frequency high amplitude bouncing), or when the heavy buffet is accompanied by a substantial deterioration of the airplanes weapon system or pilot performance.

Transonic high angle of attack buffeting is especially important in defining maneuvering limits. This type of buffeting is caused by separated flows induced by shock wave-boundary layer interactions. Buffet loads generally increase in severity as Mach number and angle of attack increase. Shock waves become stronger and the boundary layer is separated over larger areas of the wing, with more intense pressure fluctuations in the area of separated flow. Separated flows extending into the wake produce pressure divergence at the trailing edge which causes fluctuations in circulation and wing lift. The combination of these unsteady pressure fluctuations, or buffet loads, cause a dynamic response or buffeting of the structure which interacts with the natural structural modes and is transmitted throughout the aircraft. Vibration levels at the pilot's seat are, therefore, affected by: the strength of the fluctuating buffet loads, the response of the structure, and the location of the pilot's seat in relation to the natural structural modes. In addition to their effect onthe elastic structure of the aircraft, the buffet forces can also cause rigid body motions of the aircraft and induce stability and control problems. These may be coupled with the structural vibrations or with control-system-induced oscillations.

The detailed review of flow fields and separated boundary layers in Chapter 3 of AR-82 and the extensive coverage of wing and tail buffet in Chapter 7 of AR-82 will provide a more complete insight into the causes and nature of buffet. Since weapon bays on some aircraft designs must be open during transonic flight, a review of weapon bay buffeting is also covered in Chapter 7 of AR-82.

The usual criteria to show the buffet-limits are given as lift coefficients or angles of attack for buffet-onset and moderate or heavy buffet as a function of Mach number.

Correlation of the buffet-onset data in the form of lift coefficient for buffet-onset as a function of Mach number is shown in Figure 2-23 for subsonic and supersonic aircraft without maneuvering flaps. It can be seen that, before reaching the lowest value of $C_{L_{BO}}$ near the critical Mach number the values and trends of the data are similar for the two types of aircraft. In general, the lift coefficient at buffet onset decreases steadily with increasing Mach number ($M \leq 0.8$). In the case of the subsonic aircraft, the values are closer together than for the supersonic aircraft, and the slope with the Mach number seems to be a little higher, though the lowest values are within the same order of magnitude. The value of $C_{L_{BO}}$ can be raised significantly for the lower Mach number mainly by leading edge devices and even more by leading and trailing-edge maneuver flaps.

Despite considerable effort to find a reasonable correlation between some of the geometric parameters and the buffet-onset criteria, only some general tendencies could be found. As can be seen from Figure 2-24 there seems to be a tendency of increasing the angle of attack at buffet-onset α_{BO} with the wing-sweep, which is stronger for the subsonic aircraft. Nevertheless, for the lift coefficients at buffet-onset C_{LBO} this tendency could not be found. In general, it can be concluded that for the available data there are no apparent correlations between C_{LBO} and one single geometrical parameter.

Within the classified version of this report, a correlation of the available data of the two aircraft with variable geometry wings is shown. It should be noted that for those configurations, increasing sweepback changes the effective aspect ratio and the taper ratio of the wing. The effect of the sweep back depends strongly on the Mach number. There is an optimum, which is reduced by increasing the Mach number and shifted to higher sweep back angles with increasing Mach number. The effect of aspect ratio was also analyzed, and there is also an optimum which is reduced by increasing Mach number and decreasing aspect ratio.

To find a reasonable correlation between the buffet-onset criteria and combinations of geometrical parameters of the non-variable sweep wing aircraft without leading edge devices, nearly all reasonable possibilities were examined without success. The parameter:

$$\frac{C_{LBO}}{A\,.\,\lambda} = f\,(t/c)$$

defined as the buffet-onset lift coefficient divided by the aspect ratio and taper ratio as a function of the medium wing thickness ratio seems to be of some importance as shown in Figure 2-25.

For the heavy or moderate buffet, few data are available for analysis. An examination of the difference between the angle of attack at buffet-onset andheavy buffet shows two tendencies:

- the difference in angle of attack (and therefore in lift coefficient), is increased by maneuvering flaps

- modern aircraft designed with leading edge extensions (strakes) seem to improve buffet to such a large extent that transonic heavy buffet does not seem to be a maneuver limitation

CHAPTER 3

DESIRABLE APPROACH TO DETERMINE MANEUVER BOUNDARIES AT THE DESIGN STAGE

3.0 INTRODUCTION

Superior maneuverability is a fundamental performance requirement in the close-in air-to-air and air-to-ground combat environment. The maximum turn rate that the fighter pilot demands for the majority of the combat duration, results in high angles of attack and consequently separated flow conditions (on the wing and sometimes on other parts of the aircraft), and finally in inevitable performance and flying quality degradations. In addition, the new generation of highly maneuverable fighters, resulting from recent advances in aerodynamics, propulsion, structures and control system technology, with their enhanced high angle of attack capability in the subsonic, transonic and even in the supersonic flight regime, has significantly enlarged the flight envelope where angle of attack limitation is a primary consideration. Thus, designing for optimal maneuver characteristics, appears as an important prerequisite from the conceptual to the final design stage, challenging the designer's ability to determine the aircraft maneuver boundaries with the best possible accuracy. Unfortunately, this is a difficult task, because the complexity of the phenomena involved in these high angle of attack/separated flow conditions prevents the use of expeditive purely theoretical prediction methods. Moreover, methods based on wind tunnel testing at high angles of attack, particularly in the transonic range, also present difficulties of various natures, due to the sensitivity of the separated and unsteady flow to Mach and Reynolds number effects, to tunnel turbulence, noise and to model structural interference. Therefore, an appropriate combination of theoretical and experimental methods and correlations with flight test results on similar aircraft are needed to determine these limits with an acceptable accuracy.

The physical phenomena determining aircraft maneuverability limits are described in Sections 2.1 to 2.5 of this Report and are discussed in considerable detail in other AGARD publications. The present Section will present design procedures and their use to establish the maneuver boundaries. After a brief review of the principal phenomena from the pilot and engineering standpoint, the procedures available to the engineer to determine and optimize the aircraft maneuver boundaries are presented.

First, it is assumed that for each candidate configuration selected, the aircraft basic design characteristics as the weight, wing area, flap system, thrust etc... were determined by performance considerations, including the maneuverability requirements. From strictly a performance point of view, aircraft maneuverability is determined by the following parameters:

- wing loading
- thrust loading
- maximum usable lift coefficient vs Mach number
- lift-drag polar curves

3.1 *MANEUVER BOUNDARIES - THE PILOT'S VIEWPOINT*

The maneuver boundaries will first be defined as viewed by the fighter pilot. In a tactical combat environment, the pilot requires the maximum turining performance, taking into account, however, various constraints depending on the particular mission situation. At the beginning of an offensive phase of an air-to-air combat, gross maneuvers will be carried out in order to acquire a firing position, followed by a more or less precise tracking phase. For instance, to release an advanced highly maneuverable missile it will be sufficient to keep the target within a cone of about 20° of half angle, whereas for a spot-harmonized gun the angle must be about ± 2 milliradians. Consequently, to satisfy the more precise gun aiming requirement, the usable angle of attack will be limited by a lower acceptable flying quality degradation, and the usable turn rate for gun firing will be lower than for missiles. In a defensive posture, the pilot will normally produce a tighter turn rate than in the offensive situation. At low altitude, the turn rate will be limited by the airplane sustained turn performance, but at higher altitudes the tightening of the turn will be stopped only at the edge of a full loss of control. These considerations apply equally to the air-to-ground combat situations with some differences resulting from the requirements of this particular environment (e.g. SAM and AAA threat).

The previous discussion indicates that the maneuver boundaries are essentially task-dependent and a precise definition covering all the cases is not possible. Therefore, as an acceptable simplification two extreme maneuver boundaries are proposed as viewed by the fighter pilot:

- an offensive combat maneuver boundary for precision tracking (the weapon aiming maneuver limit)
- a defensive combat maneuver boundary where the pilot requires the maximum practicable instantaneous turn rate, determined by the maximum usable performance (maximum usable lift) or by a serious stability deficiency and (or) a full loss of control.

The basic task of the designer in the conceptual design phase will be to select and optimise configurations able to meet the program specifications, in particular the maneuverability requirements. A first difficulty to be faced is that there are no exact engineering equivalencies to the previously defined "fighter pilot maneuver boundaries". The problem can be dealt with in two phases:

(1) the various "engineering maneuver boundaries" will be established using proven design criteria.

(2) from these limits the "fighter pilot maneuver boundaries" will be identified by synthesis based on past experience and on semi-empirical procedures.

It is worth mentioning that the maneuver boundaries depend critically on the airplane configuration, in particular on the external stores carried. In the subsequent Sections the clean airplane configuration will be considered, or the airplane equipped with small, short range air-to-air missiles. The maneuverability boundaries of heavily loaded ground attack fighter-bombers can be studied by similar methods, but this problem will not be addressed. In addition topics related to engine operation at high angles of attack will be not discussed.

3.2 MANEUVER BOUNDARIES - THE ENGINEER'S VIEWPOINT

Engineering maneuver boundaries will be presented on an α, Mach diagram where in addition constant C_L curves are also included as shown in Figure 3-1. Section 2 of this report has already reviewed qualitatively the limits encountered when increasing angle of attack at constant Mach number, and the engineers task is to establish the relationship between the engineer and pilot standpoints (Figure 3-2).

Maneuverability characteristics are usually summarized into operational diagrams (Figures 3-3 to 3-5) displaying excess thrust contours, minimum turn radius/maximum turn rate contours and turning combat boundaries. From an examination of these diagrams, it can be concluded that trading wing loading against thrust loading is a matter of air combat tactics and that the final choice can be based on other performance requirements (as range, loiter, ceiling, etc...). On the contrary, maneuver characteristics depend essentially on maximum usable C_L.

For the designer, to obtain the highest maneuverability, he must:

- optimize the configuration for the highest $C_{L_{max}}$
- minimize characteristics which limit the usable $C_{L_{max}}$

To accomplish this difficult task in the various phases of the design development an appropriate application of a large variety of design techniques is required. The following design tools are currently used:

- theoretical methods
- semi-empirical methods and criteria
- analysis based on static and dynamic wind tunnel testing
- free flight model testing
- simulation (air-combat and spin resistance simulators)
- correlation with flight test results of similar aircraft

These techniques will now be discussed and their applications illustrated by practical examples.

3.3.1 THEORETICAL METHODS

In spite of the advances made in theoretical aerodynamics, including the transonic regime and the interaction with viscous flow (references 9, 20, and 21), applicability of these purely theoretical methods to analyze the high angle of attack combat situation is very limited. There exists, however, a prediction method for buffet onset and light buffet, (reference 19) on moderate sweepback aspect ratio aircraft (e.g. the transonic trainer). This type of aircraft is frequently limited by buffet in precision flying in the transonic range and thus, buffet is a possible transonic maneuver limit. Generalization of these methods to the more complex three-dimensional configuration must be encouraged in view of preliminary evaluations in the conceptual design stage.

Another new topic of theoretical calculations, which will play an increased role in the design of future fighters is the attached vortex flow originated from slender wings, canards, strakes, etc... Experiments on these slender configurations have shown that the suction induced by the vortices results in a substantial lift increase, and in addition, they can generate favorable aerodynamic interactions when properly managed. A considerable effort is now underway to develop a suitable analytical procedure for predicting the development of attached vortices and their effect on the aircraft high angle of attack performance characteristics (references 9, 15, 20, 21, 22, 23).

3.3.2 SEMI-EMPIRICAL METHODS AND CRITERIA

In the conceptual and preliminary design stage where wind tunnel test results of the large number of configurations considered are not available, simple methods are required to formulate at least a first judgement on the advantages and deficiencies of various designs.

Concerning buffet onset and penetration, the previously mentioned theoretical method can be used for moderate sweep/aspect ratio, but for other configurations one must rely on experience with buffet characteristics of similar aircraft. Section 2.5 of this report deals with this problem in considerable detail. An extremely accurate buffet limit prediction is not required at this stage because these limits will be significantly improved, if needed, in the detailed developmental phase of the selected configuration(s).

The other maneuver boundaries will result from various estimated static stability and control characteristics. These estimations will use theory, Datcom or DigiDatcom outputs, proprietary information, etc...From these data, the well known basic stability and departure criteria are established:

$$C_{m\beta},\ C_{n\beta},\ C_{l\beta}$$

$C_{n_{\beta dynamic}} = C_{n_\beta} \cos\alpha - \frac{I_z}{I_x} C_{l_\beta} \sin\alpha$ (Directional Departure Parameter) and

$\text{LCDP} = C_{n_\beta} \left(1 - \frac{C_{l_\beta}}{C_{n_\beta}} \frac{C_{n_{\delta a}}}{C_{l_{\delta a}}}\right)$ (Lateral Control Departure Parameter) vs angle of attack and (if data are available) vs Mach number.

At this stage of the design the most important point is probably to maximize the angle of attack where nose-slice occurs. This is obtained by keeping $C_{n_{\beta dynamic}}$ and LCDP positive to as high an angle of attack as possible. Reference 8, 12, 13, 18). More conservative approaches, keeping for instance C_{n_β} positive, C_{l_β} negative, $\frac{C_{n_{\delta a}}}{C_{n_{\delta a}}}$ low and positive to the highest reasonable angle of attack appear unacceptably penalizing. Pitch-up can be eliminated at this stage by adequate design and pitch down intensity limited to minimize trim drag penalties.

Satisfying the foregoing condition does not insure that all the design objectives will be met, but these characteristics will constitute a good initial condition for the subsequent iterations.

The next step will normally consist of wind tunnel testing of cheap and simple models. At this preliminary design stage, it is recommended to build and test such models as early as practical to verify the main performance, stability and control characteristics and to correct gross deficiencies, if necessary, by fundamental design changes. It is the right time for this type of action; basic modifications decided later would be very penalizing and detrimental to the program. The tests, including compressibility effects, must cover a large angle of attack and sideslip range in reasonably small increments in order to evaluate the above mentioned parameters with an acceptable accuracy. By analysis of the test results, preliminary performance, stability and control characteristics of the various candidate configurations are obtained and the configuration(s) selected in view of detailed design and development.

3.3.3 ANALYSIS BASED ON STATIC WIND TUNNEL TESTING

After configuration selection for detailed design and development, new wind tunnel models are constructed for comprehensive testing over the entire operating range of the airplane. In addition to conventional force and moment measurements, some of these models will be specially instrumented with steady and fluctuating pressure transducers, wing-root (and other) strain-gauges, accelerometers, etc...Analysis of test results of these more sophisticated models will give new and more accurate aircraft maneuver boundaries:

(1) Buffet limits (onset and penetration)

These limits can be determined by:

- use of successive nonlinearities of the lift, pitching moment and axial force vs angle of attack curves; all these criteria indicating onset and development of separation (kinkology). (Figure 3-6, references 14, 15).
- trailing edge pressure divergence measurements at characteristic points along the wing span.
- measuring the magnitude of the oscillatory output of a wing root strain-gauge, a tip-accelerometer or a rolling-moment balance.

These methods give generally good indications as to buffet onset, but only qualitative data on buffet level and buffet development. Three methods have been proposed to deal with this last problem:

- testing of dynamically similar flexible models with elaborate instrumentation (reference 10)
- use of semi-rigid models and derivation of approximate dimensionless buffeting coefficients (same reference)
- static and oscillatory pressure measurements on rigid models and analytical determination of the aircraft response from the pressure measurements used as input (same reference)

In principle, the first method is more accurate, but in addition to being very expensive and applicable only at later stages of development when the aircraft is structurally defined, there remain some technical uncertainties (e.g. damping) which inhibit the measurement of sufficiently accurate data to justify the efforts required by this technique. On the other hand, the second method (much simpler for practical applications), was fairly acceptable in the past, particularly when comparisons were available with similar aircraft. The third method was also applied, and gave consistent results with flight test data. More details are given in chapter 7 and 8 of reference 15. Finally, it is important to point out that buffet development can be efficiently controlled by devices such as slats, fences, notches, vortex generators, strakes, canards, etc.

(2) Maneuver limits resulting from static stability and control degradation.

After selection of one or more candidates for configuration development, static stability and control characteristics will be determined by wind tunnel measurements over a broad angle of attack and sideslip range. From the results, local values of $C_{n_{\beta dynamic}}$ and LCDP are computed in the entire angle of attack and sideslip range, and by applying the departure criteria approximate maneuver limits will be obtained (assuming that longitudinal characteristics are not critical). However, it should be noted that these parameters are useful indicators only when β is relatively small and when the rolling and yawing moments

at zero β are also small. When large lateral-directional asymmetries exist at zero sideslip, as frequently happens with long pointed bodies or at large values of β, more elaborate static stability parameters are required, taking into account the strong coupling (aerodynamic and kinematic) between longitudinal and lateral motions (reference 12) (Figures 3-7 and 3-8).

Measurement of control surface efficiencies in the whole control surface deflection range, at realistic Reynolds numbers, is also very important for the following reasons.

- control effectiveness determines whether the pilot can recover from departure or other marginal situations (as post-stall gyrations)
- large control power is needed for stabilization of unstable CCV configurations and maximum available control power can be a limiting factor for maneuver boundaries of this type of aircraft
- in some cases (reference 8) high angle of attack behavior can be improved by incorporation of an artificial control limiting system
- an accurate knowledge of the control characteristics over a large α, β range is also needed for the design of departure and spin prevention systems.

In conclusion, before any dynamic measurements are made, comprehensive static stability and control measurements are in order; and due to the strong nonlinearities and rapid variations encountered in certain angle of attack and sideslip ranges the tests must be carried out with small increments of α and β, or preferably, with continuous recording. Analysis of these measurements by the previously discussed methods will result in good approximation of the maneuver limits. Additional tests and analysis procedures to determine more definite boundaries and to describe the aircraft behavior on both sides of the boundaries are presented in the subsequent sections.

3.3.4 ANALYSIS BASED ON STATIC AND DYNAMIC WIND TUNNEL TESTING

In the detailed design stage, the static wind tunnel tests will be supplemented by dynamic tests and the complete set of data will be used as input to six degrees of freedom analysis. The nature of the dynamic tests depends on the extent of the analytical objectives. They can be limited to the study of the airplane behavior below the potential maneuver boundaries, or also include the analysis of phenomena encountered beyond the boundaries, including: stall, departure, incipient spin, developed spin, and the corresponding recovery procedures. These dynamic tests are also needed to validate the static criteria and to define the maneuver limit modifications when the static criteria are not sufficient.

The most commonly used dynamic wind-tunnel testing procedure is the forced oscillation method, where the model is sting mounted on an oscillating internal balance. At high angles of attack there are some special problems as: sting interference with the model (reference 12), nonlinearity with amplitude (references 12, 28), variability with frequency (references 12, 28), which can be dealt with only on a case by case basis.

Another method sometimes used is the curved or rotating flow technique to determine the dynamic derivatives in quasi-static conditions. The results of each of these methods-or better-their combinations, can be used to compute airplane response at high angles of attack to various extreme but realistic control inputs, to uncover and if possible correct stability and control deficiencies (in particular those which limit the maneuverability). Finally, using the data acquired in the configuration development and finalization the departure and spin resistance will be analytically established.

To continue further the analysis to the spin area, the high angle of attack static and dynamic test results must be combined with rotary balance test data. The aircraft response, based on six degrees of freedom nonlinear analysis using these data, will provide the best analytical information on the incipient spin behavior, on the control actions required to recover, on developed spin modes and on recovery procedures from developed spins.

To obtain the required rotary balance data at high Reynolds numbers and to check the sensitivity of the rotary aerodynamic parameters to scale effects, a new rotary balance apparatus was recently developed for use in a large pressure tunnel at the NASA-Ames Research Center with a rotation rate able to simulate full scale spin motions (reference 8). It is expected that the data obtained with this equipment will produce improved basis for the analytical determination of future airplanes high angle of attack behavior, including their spin characteristics.

3.3.5 FREE-FLIGHT MODEL TESTING TECHNIQUES

Free-flight model testing is an experimental method not requiring in principle previous knowledge of the aircraft aerodynamic characteristics. Except when it is carried out at large scale (a very expensive technique) Reynolds numbers are low and correlation with full scale data can be questionable. However, in flight conditions not too sensitive to scale effect, or when it was possible to simulate high Reynolds number effects by some fixes, good correlation has been obtained with flight test results in many cases.

Free flight testing techniques, used in the various stages of aircraft development, are:

- Tethered model testing in wind-tunnel (remotely controlled and augmented)(Figure 3-9).
- Catapulted, control augmented model testing (Figure 3-10).
- Spin testing in vertical tunnel (Figure 3-11).
- Helicopter drop model testing (programmed or remotely operated controls (Figure 3-12).

- Large scale dynamically similar models, released from aircraft at high altitude (RPRV technique) and maneuver from a ground station (Figure 3-13).
- Small scale powered models remotely operated from takeoff to landing and carrying out complete flight test programs (e.g. high angle of attack maneuvers including spins and recovery).

These techniques, separately or in combination, can supplement other procedures of aircraft development and validate in a broad sense the results obtained.

3.3.6 SIMULATION OF AIR COMBAT MANEUVERS (air combat and spin-resistance simulators)

In theprevious techniques discussed, engineering considerations dominated without the pilot in the loop (except in some free-flight techniques, where however due to the contracted time-scale the simulation environment is not entirely realistic from the pilot standpoint). In order to correct this undesirable situation and obtain quantitative pilot evaluation data of the whole man-machine system, simulators are used to evaluate the high angle of attack characteristics with pilot in-the-loop in realistic combat situations. The basic ingredients of a proper simulation are:

- a comprehensive aerodynamic data base in an adequate α, β, and Mach number range, including all significant nonlinearities.
- an accurate mathematical model of the control systems (or a real physical representation of part of the control system, if useful)
- a realistic display of the cockpit and combat environment (visibility, instruments, force feel, visual display).

More or less sophisticated simulators are used following the design advancement and data availability (references 8, 12, 26, 27, 28).

- Single cockpit fixed-base simulators
- Single cockpit moving-base simulators, large angular and linear motions, computer controlled interactive opponents (Figure 3-14).
- Dual cockpit fixed-base combat simulators (Figure 3-15).

With the pilot in-the-loop, it becomes possible to check and refine the high angle of attack characteristics, departure and spin resistance and recovery by carrying out realistic and extreme combat maneuvers. In addition, a great deal of control system development and optimization can be effected near the end of the design cycle which will contribute to a more accurate definition of the maneuver boundaries. Two examples illustrate this important point:

(1) <u>Weapon aiming maneuver boundary refinement</u>:

As previously discussed this boundary depends principally on some basic aerodynamic characteristics. However, the control system design parameters have also a significant effect onthe aiming accuracy and their proper adaptation can require simulation exercises with tracking tasks and judgment based on pilot ratings.

(2) <u>Automatic maneuver limitations</u>:

Low altitude-high speed maneuver boundaries result from structural limitations, which due to the rapid flow changes in the transonic range present a complex pattern. In other flight conditions the aircraft limitations are generally altitude and Mach number dependent. To be acceptable to the pilot as Flight Manual limitations they must be sufficiently simple.

Anothertechnique to deal with this problem on an aircraft equipped with a full-authority stability and control augmentation system is to incorporate automatic limitations through the control system design. The best way to define these boundaries in the whole flight envelope is to establish a complete mathematical model of the aircraft, including the control system and structural interfacing, and to carry out ultimate maneuvers on the simulator. Control power and control surface deflection rates are defined in the procedure and their efficiency on the limitations checked. All the new-generation highly-maneuverable fighters are incorporating this type of automatic limiting system.

3.3.7 CORRELATION WITH FLIGHT TEST RESULTS OF SIMILAR AIRCRAFT

In the previous sections, methods to establish maneuverboundaries of combat aircraft were discussed and it was shown that powerful tools exist to define these boundaries with a reasonable accuracy. The method presented in this section is applicable mainly in an evolutionary design technique frequently practiced or to derivative aircraft from a basic configuration. Qualitatively, it can be used in many cases, considering that present combat aircraft designs normally belong to four classes of wing design:

(1) low sweepback/high aspect ratio wing

(2) moderate sweepback/aspect ratio wing

(3) slender wing

(4) variable geometry

In configuration development - at least in conceptual and preliminary design - available flight test data for a similar class of airplane can be a valuable data base. Differences existing between a new design and a design fromthe same class can be evaluated by semi-empirical methods or tests and combined with the existing flight test data. The method was successfully applied in the past; however, with new unconventional aerodynamic shapes which are extremely configuration dependent care must be exercised, and for these cases detailed application of the methods discussed in the previous sections is recommended.

3.3.8 CONCLUSIONS

Maneuver boundaries as viewed by the fighter pilot can be correlated with the various engineering boundaries and the operational limitations of a combat airplane can be defined with an increasing accuracy as the design is developed.

Figures 3-16(a) and 3-16(b) show such correlations between predictions based on wind tunnel test analysis and flight tests for two different aircraft. Figure 3-16(a) is for an airplane with a moderate sweepback; figure 3-16(b) is for a high sweep case of a variable geometry aircraft.

The limiting phenomena depend on the tactical combat environment. Two types of operational boundaries are an offensive maneuver boundary for precision tracking, representing the weapon aiming maneuver limit; and a defensive escaping combat maneuver boundary where the pilot will use the maximum performance until experiencing a significant degradation of stability and (or) control. These boundaries result from phenomena such as heavy buffeting, wing rock, wing drop, nose slice, pitch-up, etc.

Design procedures to determine these limits and the corresponding operational maneuver boundaries were discussed. The procedures applicable at the various design development stages include analytical and semi-empirical methods, and more or less sophisticated testing and simulation techniques.

CONCLUSIONS

Combat aircraft designs are evolved in an attempt to best meet a combination of desired military aircraft requirements and constraints such as range, speed, safety, weapon load, operational field length, inflight turn capability, procurement cost and operational cost. This results in aircraft that are satisfactory within a particular flight envelope but whose maneuverability is limited by various departure or physical environmental phenomena at different locations in the operational flight envelope. Many modern fighter aircraft have sufficient thrust and control capability to achieve very high angles of attack (well beyond the stall) and can therefore encounter limiting phenomena that were not seen in older fighters.

The particular limiting conditions are very much aircraft configuration sensitive, and very often local area detail sensitive.

Reasonable estimates of these limitations, during the aircraft design stage, must be obtained from a rigorous dynamic analysis of flight behavior utilizing estimated full scale weight, inertia, and flexibility characteristics; coupled with wind tunnel test data of a precise model tested at the flight Mach Numbers and at reasonable test Reynolds Numbers.

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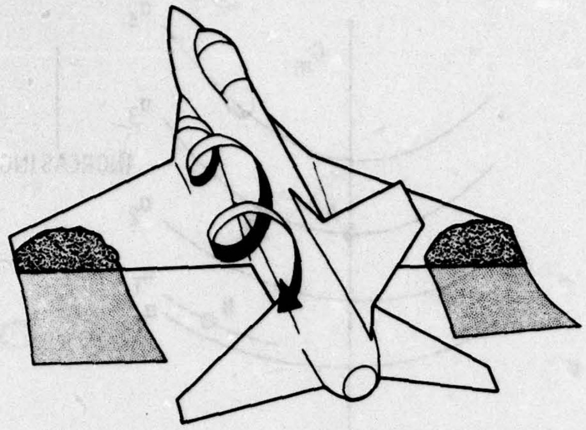


Figure 2-1.- Illustration of stalled and vortex flows at high α

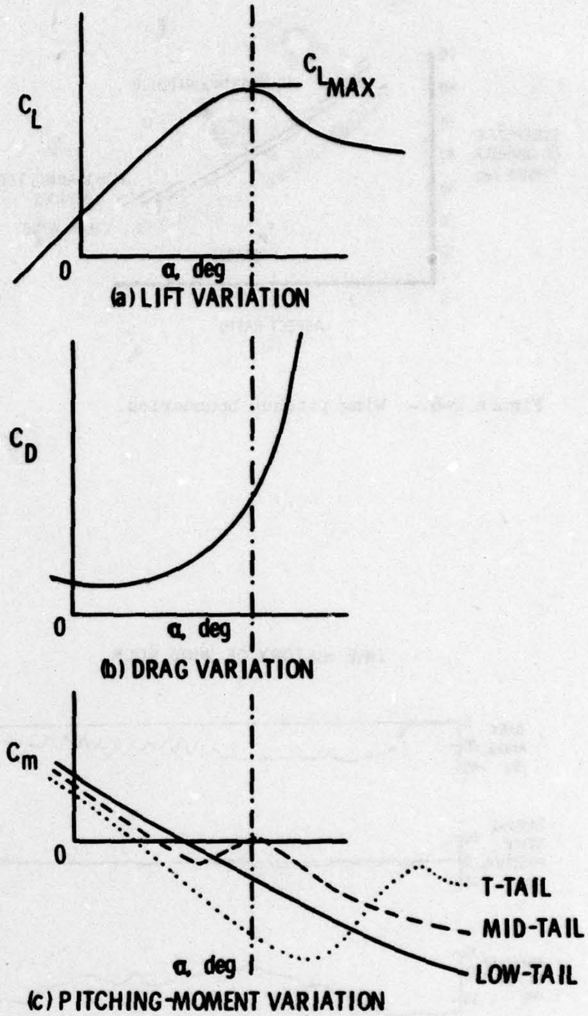


Figure 2-2.- Typical aerodynamic force and moment variations with α

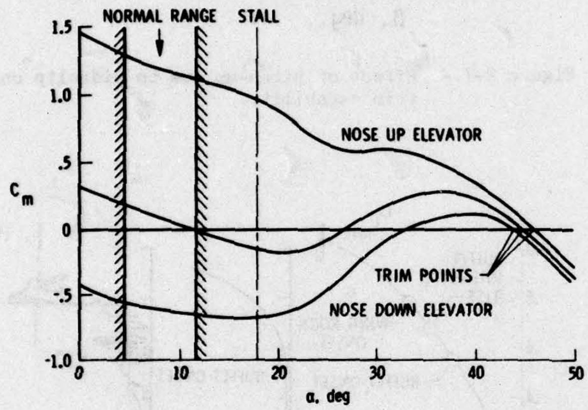


Figure 2-3.- Longitudinal aerodynamic pitching moments for a configuration with deep stall.

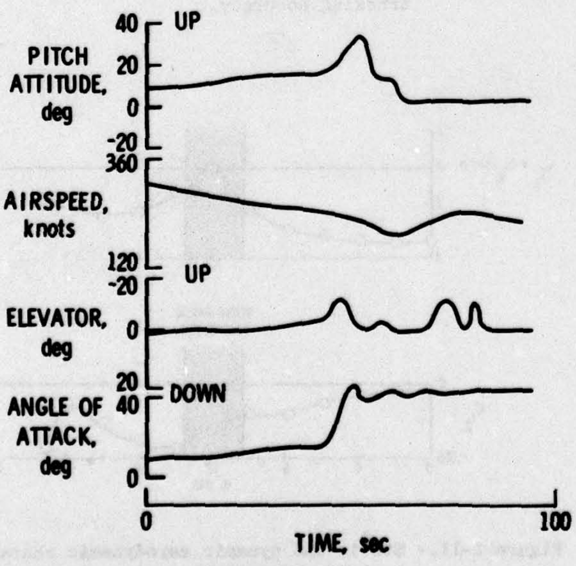


Figure 2-4.- Time histories of simulated deep stall.

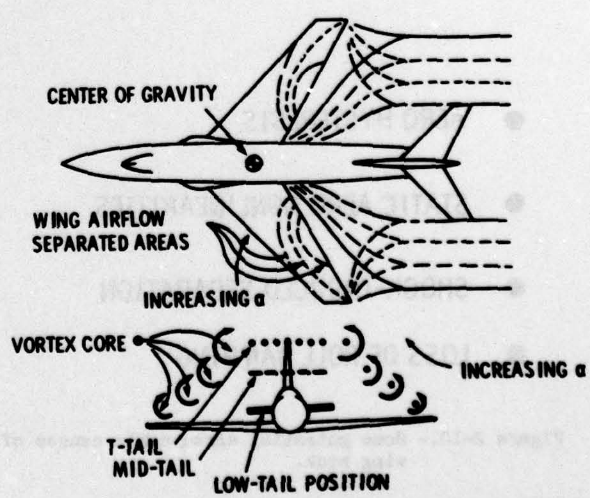


Figure 2-5.- Effect of increasing α on wing flow separation and vortex core locations.

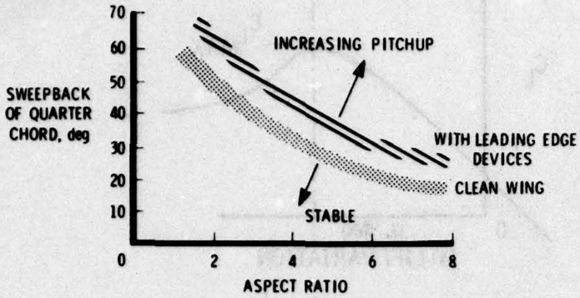


Figure 2-6.- Wing pitchup boundaries.

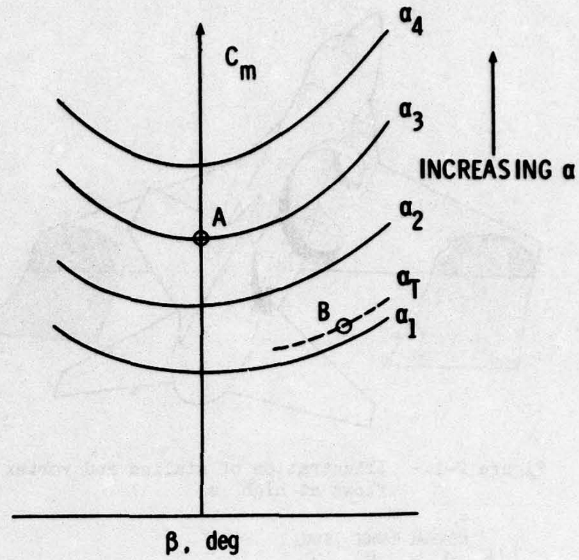


Figure 2-7.- Effect of pitch-up due to sideslip on trim capability.

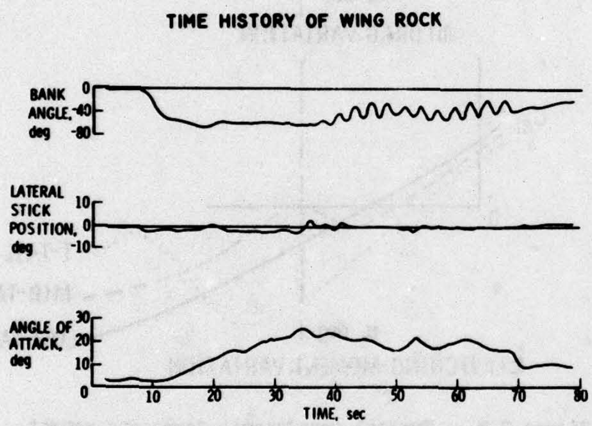


Figure 2-8.- Flight record of fighter encountering wing rock.

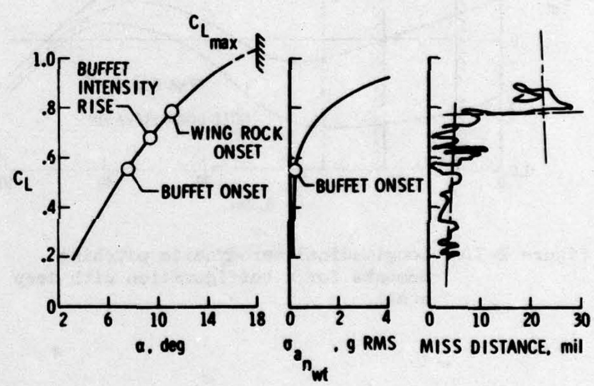


Figure 2-9.- Effects of buffet and wing rock on tracking accuracy.

- AERO HYSTERESIS
- STATIC AERO NONLINEARITIES
- SHOCK-INDUCED SEPARATION
- LOSS OF ROLL DAMPING

Figure 2-10.- Some potential aerodynamic causes of wing rock.

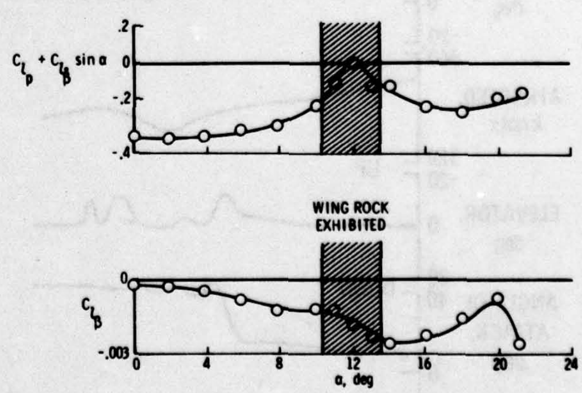
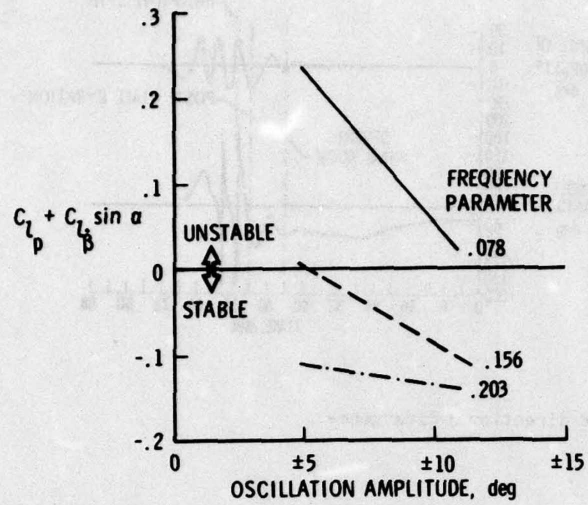


Figure 2-11.- Static and dynamic aerodynamic characteristics for fighter with wing rock.



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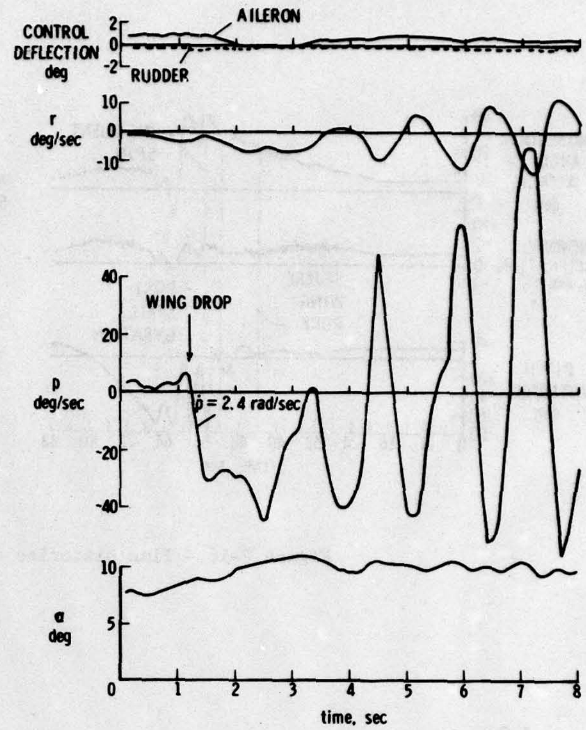


Figure 2-13.- Wing drop and wing rock. M = 0.77

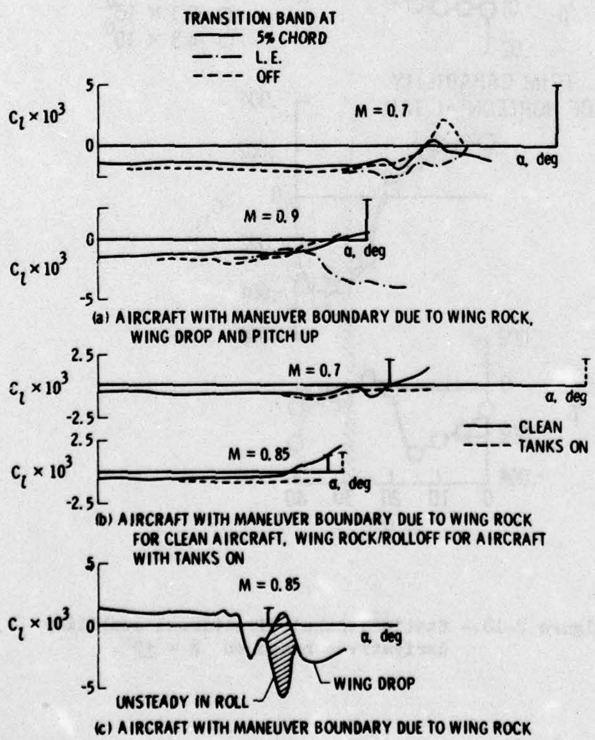


Figure 2-14.- Wind-tunnel measurements of rolling moments at $\beta = 0°$ for three airplane configurations with roll limitations.

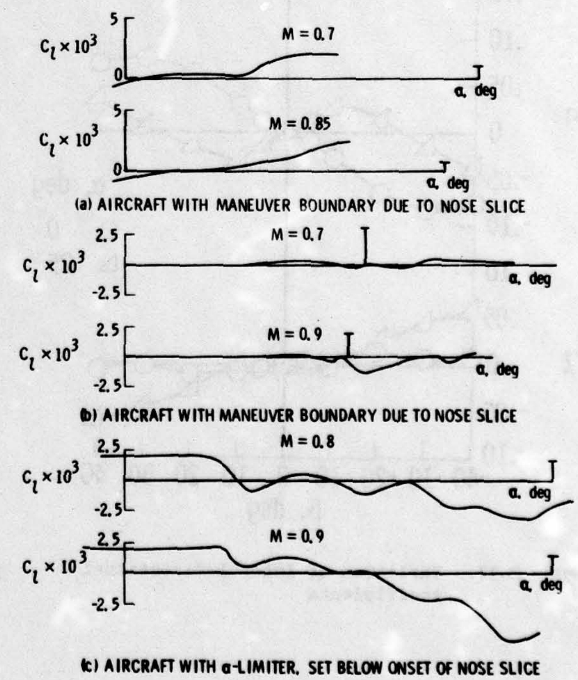


Figure 2-15.- Wind-tunnel measurements of rolling moments at $\beta = 0°$ for three airplane configurations with no roll limitations.

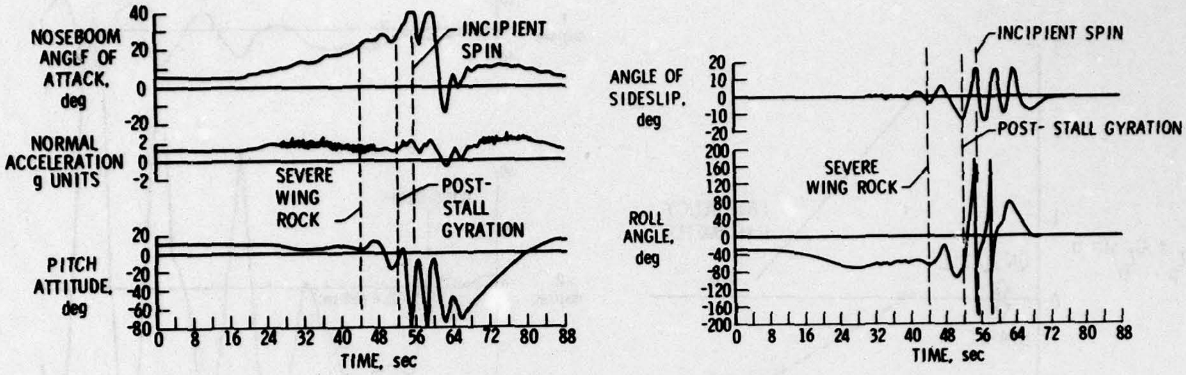
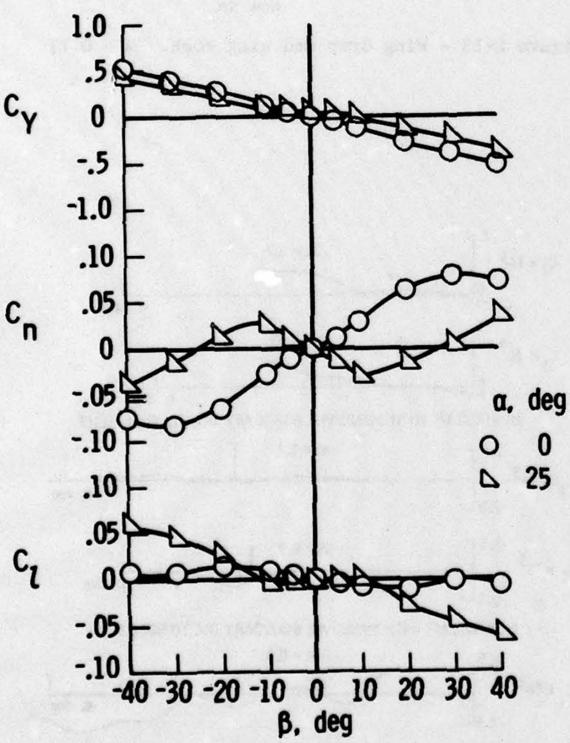


Figure 2-16.- Time histories of directional divergence



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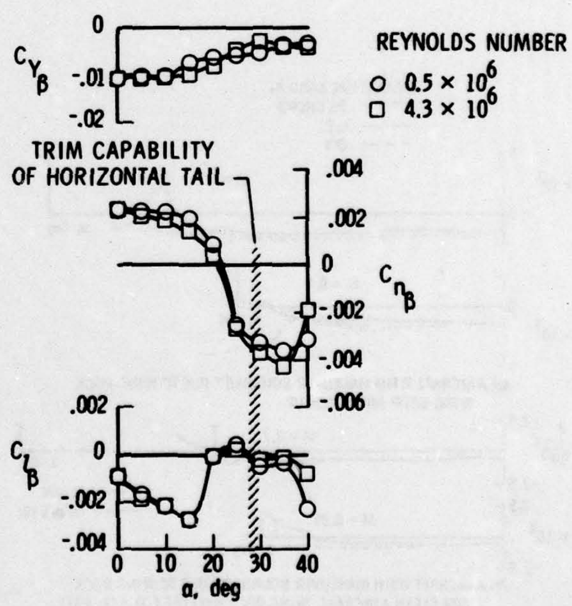

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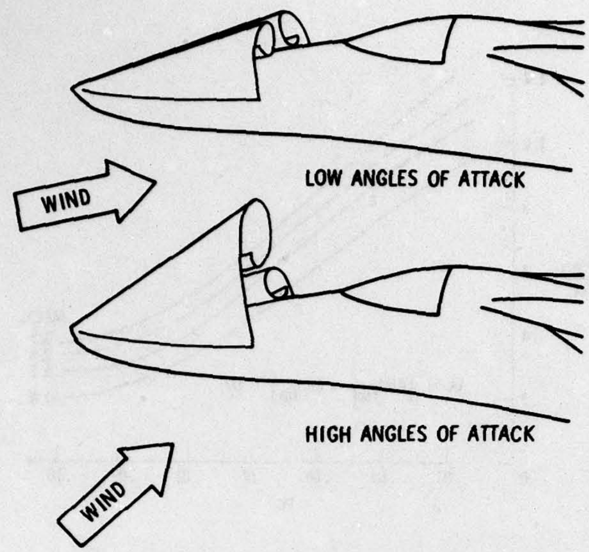

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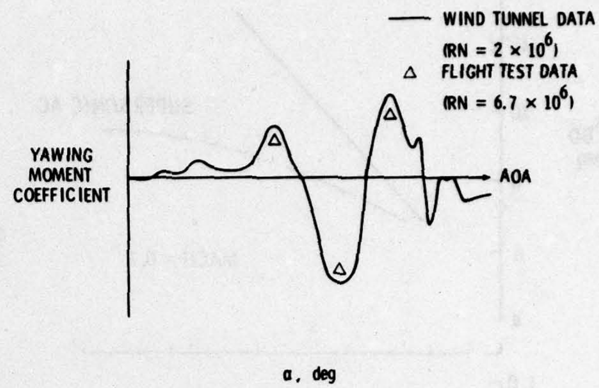

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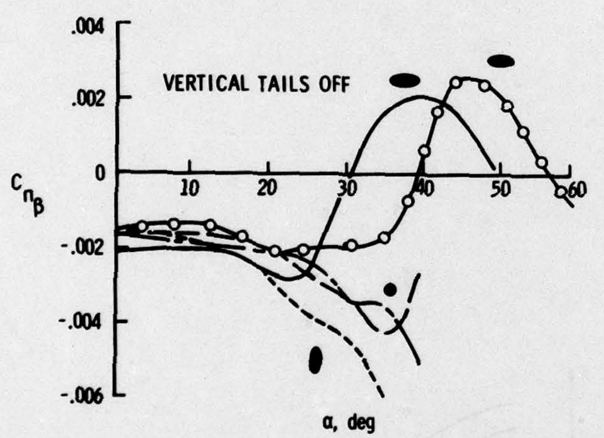

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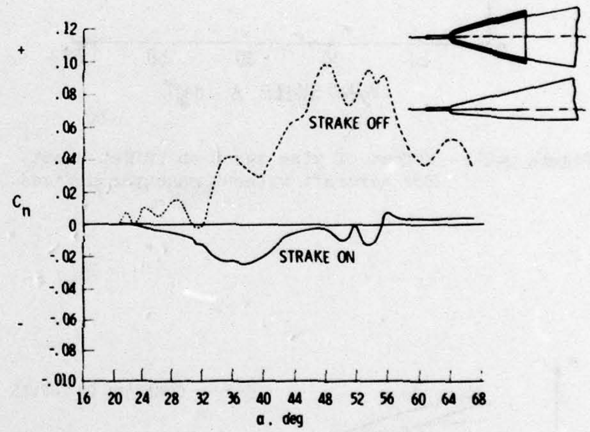

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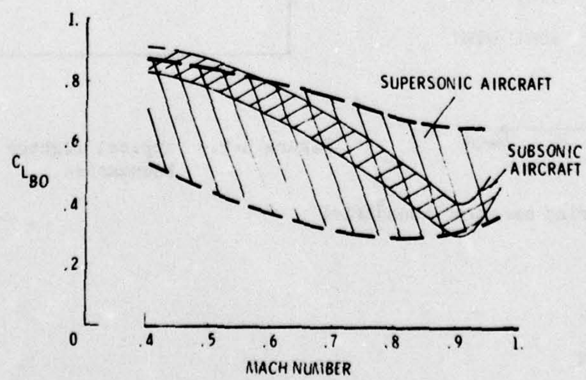

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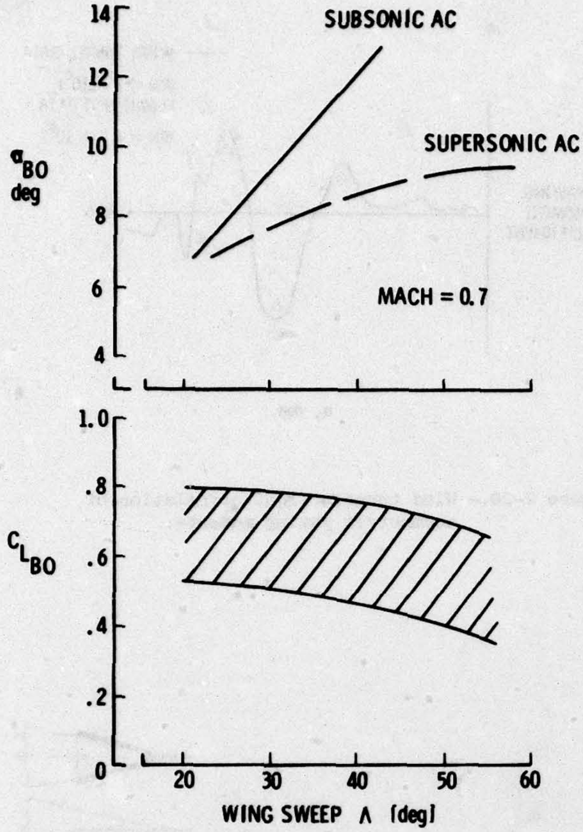

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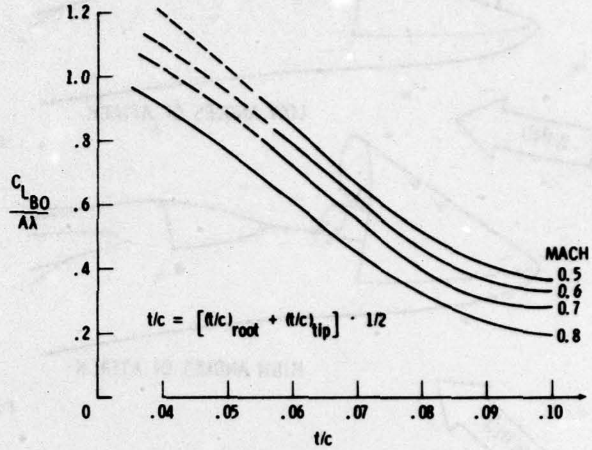

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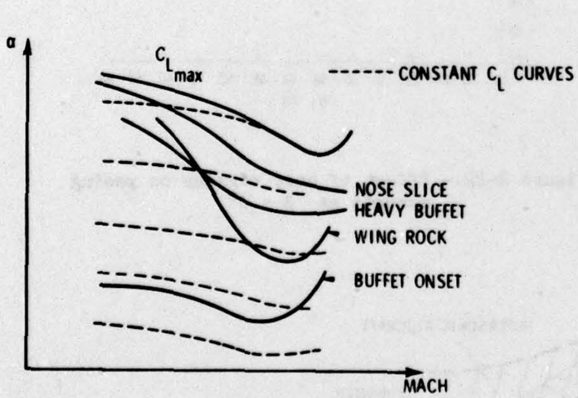
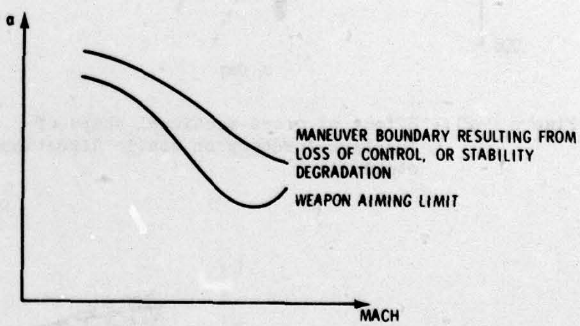


Figure 3-1.- Typical engineering maneuver boundaries



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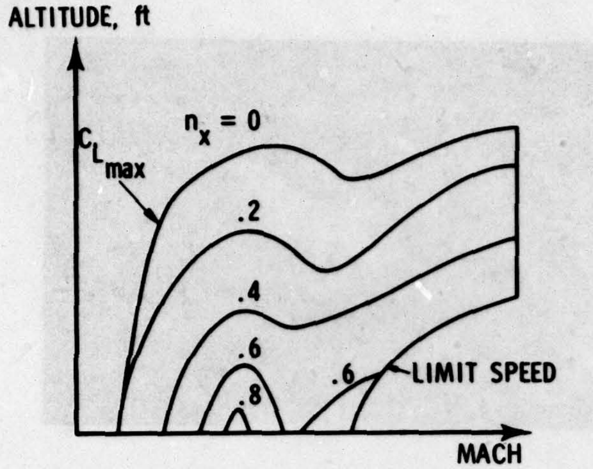

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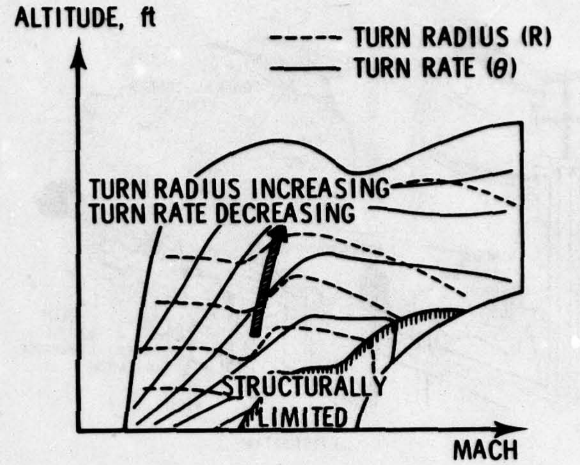

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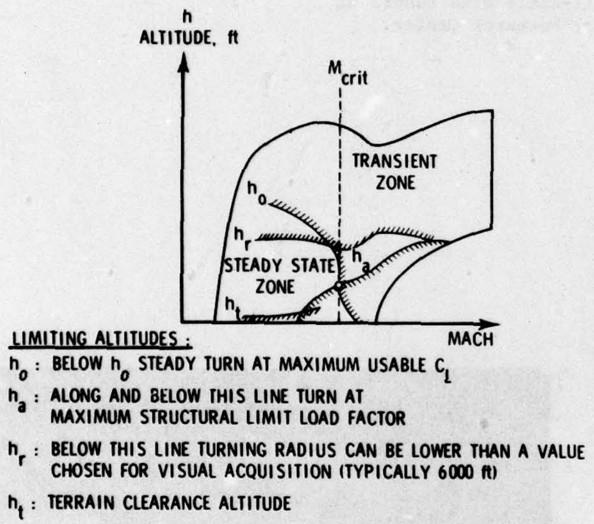

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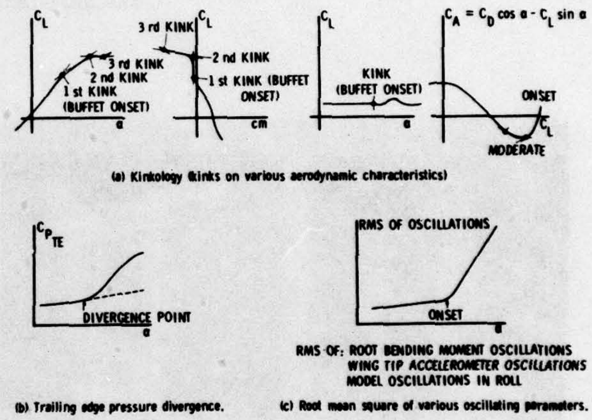

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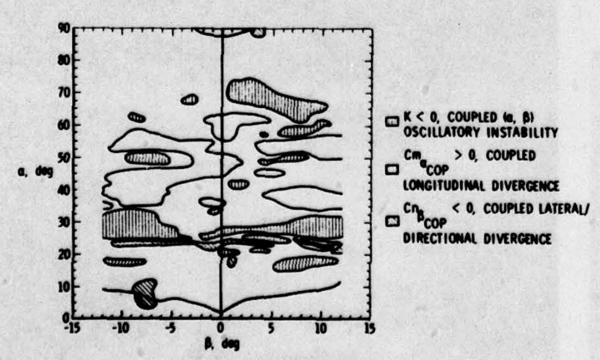

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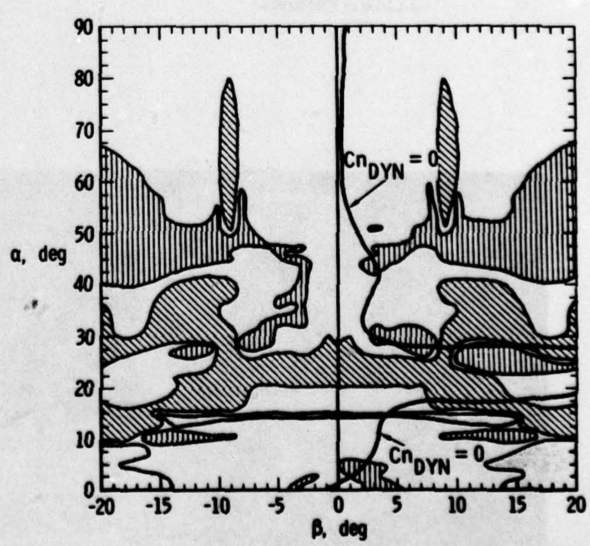

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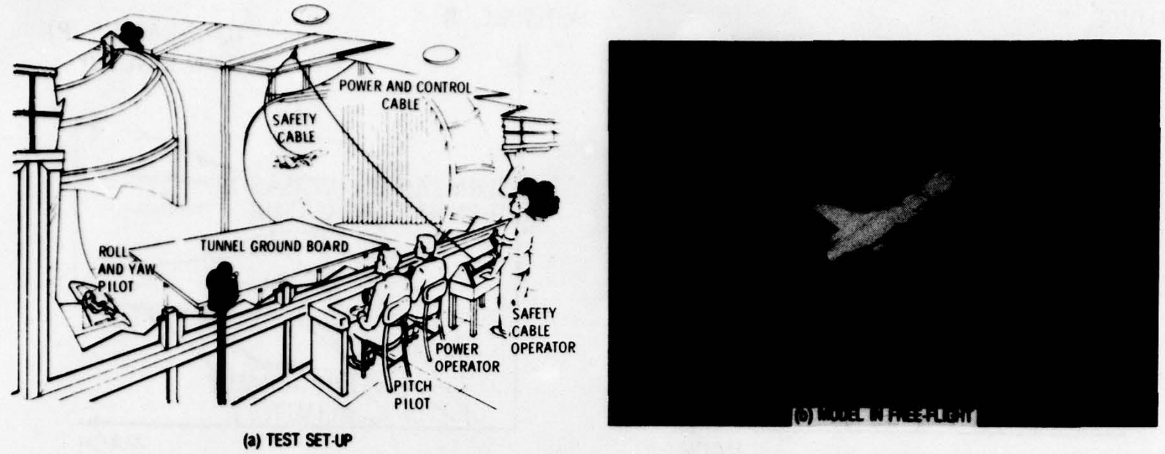

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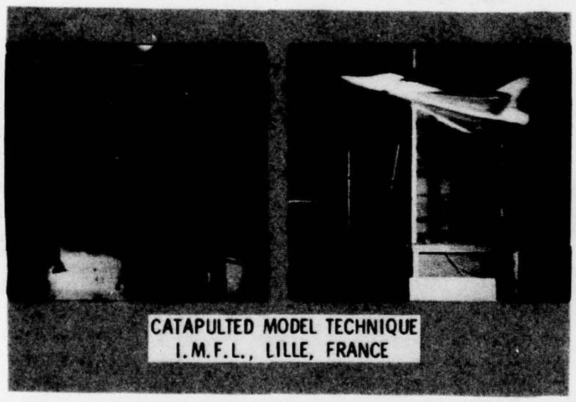

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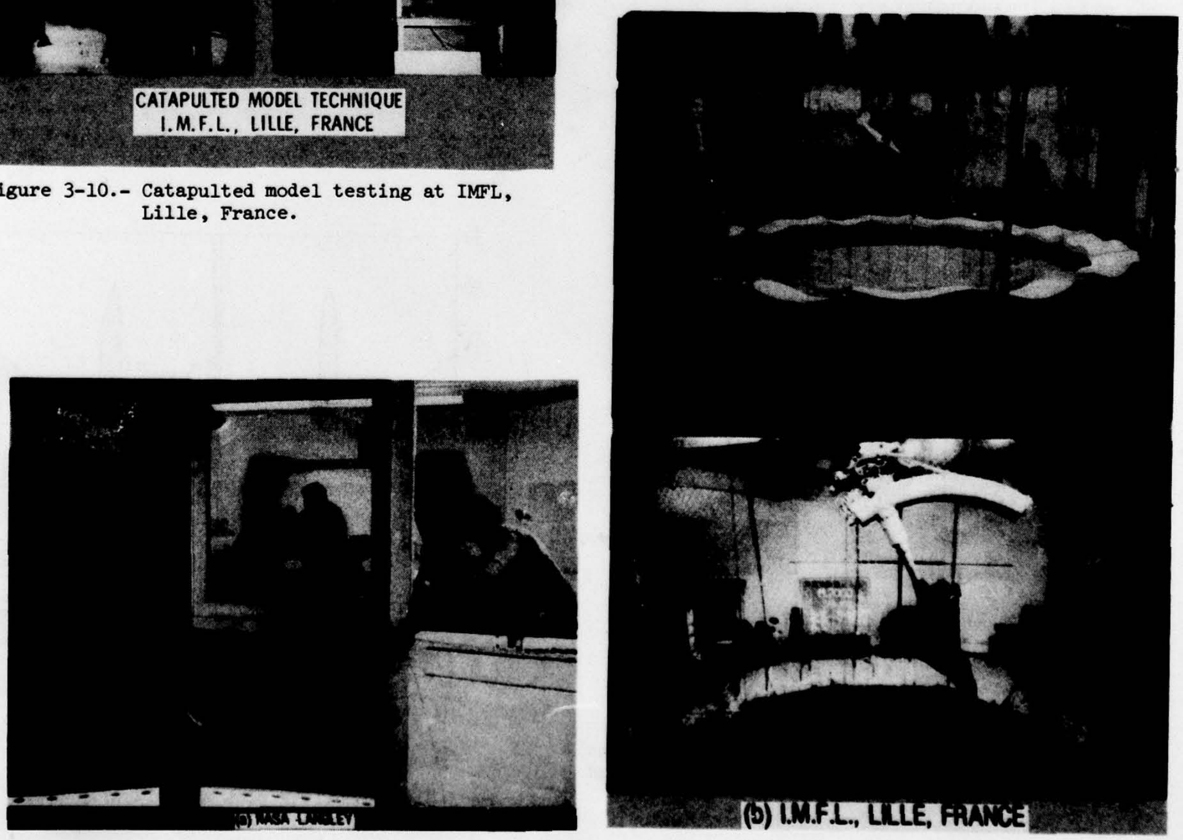

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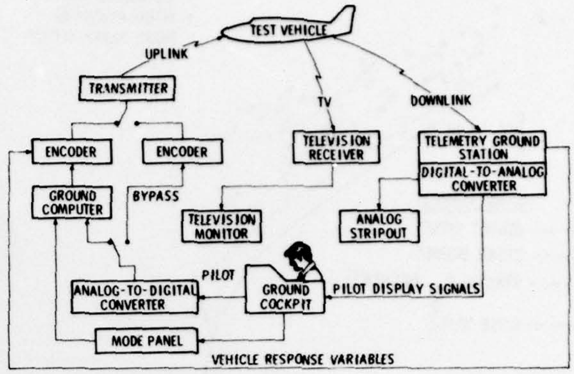
Figure 3-12.- Helicopter drop model testing (NASA-Langley)



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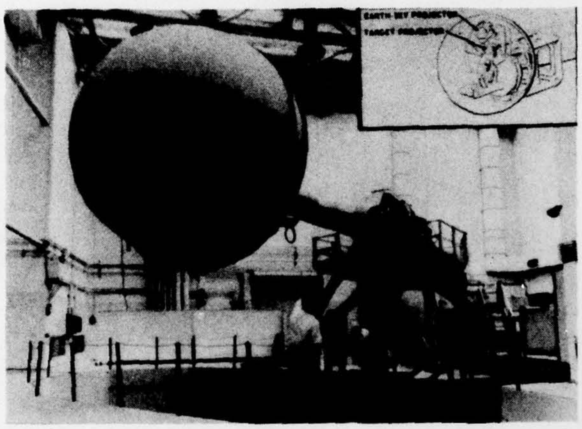

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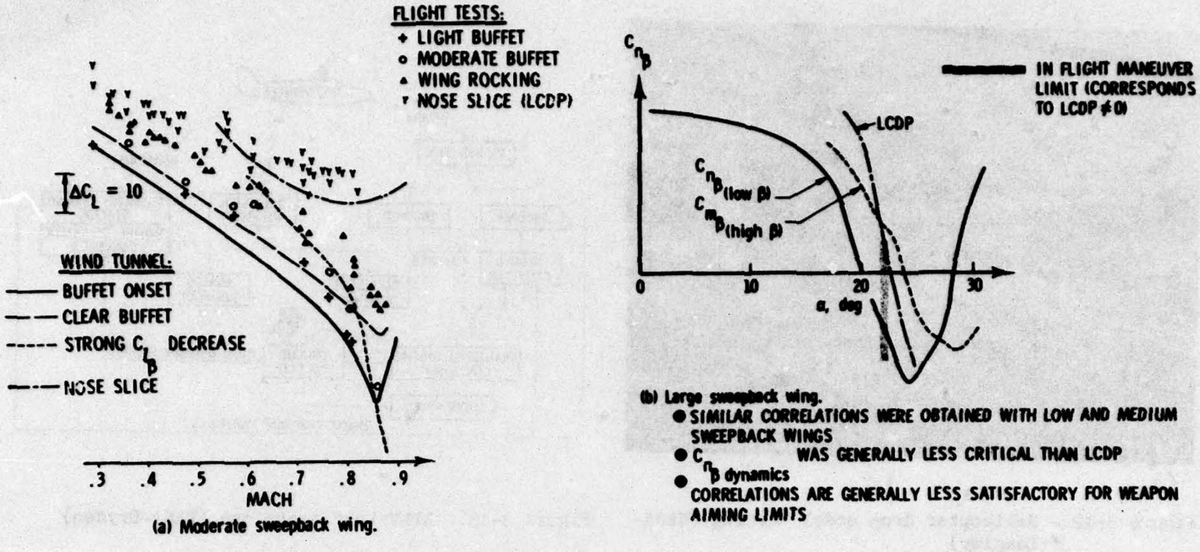

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14. Abstract

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